CERTIFICATION OF TRANSLATION

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Francis BEROGIN

METHOD OF CONTROLLING THE ATTITUDE OF SATELLITES, PARTICULARLY AGILE SATELLITES WITH A REDUCED NUMBER OF GYRODYNES

The invention relates to satellite attitude control by exchange of angular momenta delivered by a plurality of inertial actuators having rotary members mounted on the satellite platform.

The invention relates more particularly to a method and to a system for controlling the attitude of what are called agile satellites, that is to say those capable of very rapid attitude maneuvers, which are equipped with an attitude control system comprising at least two gyrodynes.

It is known that a gyrodyne, also called a gyroscopic actuator and often denoted by the acronym CMG (Control Moment Gyro), is distinguished from reaction wheels, commonly used for controlling the attitude of a satellite by exchange of angular momenta, in that the control moment gyro includes a rotor driven (by a motor) so as to rotate about a rotation shaft which is itself fastened to a support, called a gimbal, which is steerable (by at least one other motor) about at least one gimbal shaft fixed relative to the platform of the satellite, the axis of rotation of the rotor moving perpendicular to the gimbal shaft, whereas a reaction wheel is driven (by a motor) so as to rotate at a variable speed about an axis of rotation that is fixed relative to the platform of the satellite.

Agile satellite attitude control methods and systems of the prior art generally comprise a cluster of three or four control moment gyros delivering large torques along the three axes of the satellite.

One also well-known method consists in using two head-to-tail control moment gyros (their angular momenta being equal in modulus and opposed in direction) for producing torques in a direction, in this case the bisector of said angular momenta.

Moreover, patents US-5 681 012 and US-6 360 996 describe a method using two control moment gyros to produce torques along two different axes.

For this purpose, and with reference to figure 1, which shows schematically the arrangement of the two control moment gyros by the orientation of their gimbal axes and angular momentum vectors developed relative to the reference orthogonal coordinate system (X, Y, Z), the gimbal axes A1 and A2 of the two control moment gyros are mounted in the plane defined by the two axes X and Y of the coordinate system, this (X,Y) plane being orthogonal to the Z axis, which is for example the pointing axis of an instrument on board the satellite and which is intended to be tilted. The angle φ between the two gimbal axes A1 and A2 must necessarily be nonzero in order to obtain the desired effect. According to the two aforementioned US patents, the preferred angle φ is 120°C. The angular momenta $\vec{H}_1$ and $\vec{H}_2$ of the two control moment gyros are thus constrained to move in the planes P1 and P2 respectively, these being orthogonal to A1 and A2 respectively, and making between them the same angle φ. In the canonical position, the angular momenta $\vec{H}_{1can}$ and $\vec{H}_{2can}$ of the two control moment gyros are advantageously aligned in a head-to-tail configuration along the Z axis, so that the total angular momentum of the pair of control moment gyros is zero. This arrangement is called a "skewed scissor pair".

Starting from this canonical configuration, the angular momenta $\vec{H}_1$ and $\vec{H}_2$ of the control moment gyros are each pivoted about their respective gimbal axis A1 or A2 in

such a way that the resultant torque has nominally a zero component along the Z axis, without which at least a third actuator, acting along the Z axis, would have to compensate for this component, which could be high owing to the fact that the torques delivered by the control moment gyros are very high.

In order for this component along the Z axis to be zero, it is necessary to constrain the temporal movement of the rotation angles L1 and L2, given to the two control moment gyros respectively, about their respective gimbal axis A1 and A2, from the canonical position.

More precisely, according to US-5,681,012, it is necessary that:

$$\frac{dL1}{dt} \cdot \sin(L1) = \frac{dL2}{dt} \cdot \sin(L2)$$

that is to say, by integrating:
cos (L1) = cos (L2) + constant,
the constant being zero since L1 = L2 = 0 at time t = 0.

Consequently, in order for the control method according to US-5,681,012 to be able to be implemented, it is essential that the rotation angles of the control moment gyros, from their canonical position, be equal in absolute value, it being possible for the angles to have the same sign (L1 = L2) or opposite signs (L1 = -L2). The skewing of the two gimbal axes A1 and A2 with a nonzero angle φ then ensures generation of torques in two different directions U1 and U2 in the (X,Y) plane, depending on whether the signs of said rotation angles are the same or are opposed, as described in detail in US-5,681,012, to which the reader may advantageously refer for further details about this subject.

However, it is important to note that, in principle, the generation of these two torques can be accomplished, according to this known method, only sequentially and not simultaneously, as it is not possible to have L1 = L2 and L1 = -L2 at the same time.

The first consequence of this known system and known method is the noncontrollability along the three axes of the system for small angles. Other actuators must be used to overcome this drawback. In addition, to tilt the Z axis about any axis U in the (X,Y) plane, it is necessary to decompose the rotation R(U) about the U axis into a product of two rotations, the first of which takes place about the U1 axis (R(U1)) and the second about the U2 axis (R(U2)).

Thus, to generate the rotation R(U), the satellite will firstly be tilted along U1 in order to perform the rotation R(U1), then along U2 in order to perform the rotation R(U2), with a stop phase between the two rotations.

The limitations of this method are therefore noncontrollability at small angles and also considerable suboptimization in the performance of maneuvers at large angles.

Patent US-6 360 996 relates to improvements made to the method according to US-5 681 012. The basic principle, namely the skewed scissor pair configuration, is maintained. However, in addition, deviations with respect to the constraints:

$$\frac{dL1}{dt} \cdot \sin(L1) = \frac{dL2}{dt} \cdot \sin(L2)$$

that is to say L1 = L2 or L1 = -L2, are accepted in US 6360996, the disturbing torques induced along the Z axis then being compensated for by a variation in the speed of the control moment gyro rotors. Thus, complex

couplings appear between the control along the (X,Y) axes and the control along the Z axis, in particular in maneuvering mode.

These couplings are not easily manageable and they induce the risk of saturation of the actuators along the Z axis. Management of this saturation is a central feature of the method, as results from the description given in US-6 360 996, the more so as the control method described in that patent uses only very conventional tilt guidance concepts, by determination of trajectories and generation of torques to be applied to the satellite in order to perform the determined trajectories.

To alleviate the aforementioned drawbacks of the prior art (use of two control moment gyros to create torques along an axis, or along two axes, but with strong implementation constraints), the invention proposes a satellite attitude control system that comprises a pair of control moment gyros and at least a third actuator in a configuration different from those known from the prior art, in particular the patents US-5 681 012 and US-6 360 996, so as to achieve attitude control along three axes of the satellite, and also rapid tilts, with guidance and control laws that are very simple to implement, and with controlled inter-axis couplings.

For this purpose, the method according to the invention, for controlling the attitude of a satellite equipped with an attitude control system in a reference coordinate system (X, Y, Z) for positioning the satellite, and comprising at least three actuators called main actuators, two of which are control moment gyros each having a rotor driven so as to rotate about a fixed rotation axis with respect to a steerable gimbal that can be oriented about a gimbal axis perpendicular to the rotation axis of the corresponding

rotor, and stationary with respect to the satellite, is characterized in that:

- the gimbal axes of the two control moment gyros are fixed so that these gimbal axes are parallel to each other and to the Z axis, the angular momentum vectors $(\vec{H}_1, \vec{H}_2)$ of the control moment gyros therefore moving in the (X,Y) plane and making between them an angle (α) which, by definition, corresponds to a skew $\varepsilon = 180 - \alpha$ between the angular momentum vectors $(\vec{H}_1, \vec{H}_2)$ when α is different from 0° and 180°;

- in addition to the two control moment gyros, at least a third main actuator is used as a complement, delivering torques in both senses in at least one direction not lying in the (X,Y) plane, so that this third main actuator is called the Z-axis main actuator;

- a nonzero skew angle (ε) between the angular momentum vectors $(\vec{H}_1, \vec{H}_2)$ of the control moment gyros is imparted, said skew angle (ε) preferably being chosen to be small enough not to create an excessively large internal angular momentum on board the satellite but large enough to ensure controllability of the attitude control system along the three axes (X, Y, Z) without necessarily having to modify the rotation speed of the rotor of at least one of the control moment gyros;

- the kinematic and dynamic variables, which are necessary for controlling the attitude of the satellite, such as for example the attitude angles and angular velocities of the satellite along the three axes, are estimated from measurements provided by sensors used on board the satellite;

- setpoint variables, intended to allow objectives assigned to the satellite attitude control system to be achieved, such as for example the tilting and pointing along at least one of the three axes of the (X, Y, Z) coordinate system, are calculated; and

- control commands are calculated, from differences between said estimated variables and said setpoint variables, and then sent to the main actuators, these control commands being intended to

control the change in said differences over time, said control commands transmitted to the control moment gyros comprising at least commands intended to vary the orientation of their gimbal axes, such as for example gimbal angular position setpoints that have to be generated by a local position feedback control, or electric current setpoints, for currents that have to be injected into motors for orienting the gimbal axes, etc.

This method using one pair of control moment gyros in this particular configuration, in which the angular momenta change in the (X,Y) plane with a nonzero angle α, about a position not aligned head-to-tail but with a nonzero skew angle $\varepsilon = 180 - \alpha$, and also at least one third actuator for creating nonzero torques about the Z axis normal to the (X,Y) plane, is advantageous over the prior art in that it makes it possible, as described below, on the one hand, to very simply control the attitude of the satellite along the three axes (X, Y, Z) without it being necessary to modify the rotation speed of the control moment gyro rotors and, on the other hand, to easily perform rapid tilting maneuvers of the Z axis, by advantageously applying the guidance techniques in maneuvering mode that are proposed in the Applicant's patent FR 2 786 283, all this with great ease of design of the control system, in particular with regard to management of the coupling between the (X, Y, Z) axes and the design of the actuators that result therefrom.

Other advantages and features of the invention will become apparent from the description given below, by way of nonlimiting example, with reference to the appended drawings in which:

- figure 1, described above, is a schematic representation of the arrangement of two control moment gyros, represented by the orientations of the gimbal axes and of the angular momentum vectors of the control

moment gyros of an attitude control system according to the prior art;

- figures 2a and 2b show, in a manner similar to figure 1, the orientations of the gimbal axes and the angular momentum vectors of the two control moment gyros of the attitude control system according to the invention, respectively in a principle configuration, in which a third main actuator is advantageously a Z-axis reaction wheel, and in a redundant configuration in which the third main actuator comprises two reaction wheels, the rotation axes Z_1 and Z_2 of which are skewed relative to the Z axis;

- figure 3 is a schematic representation of the three main actuators, including two control moment gyros and one reaction wheel, of a satellite attitude control system according to the invention; and

- figures 4a and 4b are schematic representations, in the (X,Y) plane in which the angular momentum vectors of the two control moment gyros of the attitude control system according to the invention change, of the effect, on the total angular momentum vector $\vec{H}$, respectively of a variation by one and the same small angle of the orientation angles of the angular momentum vectors of the two control moment gyros and of variations in small opposed angles of the same two angular momentum vectors.

To implement the satellite attitude control method according to the invention, one possible, but not unique, embodiment of the control system is the following. The satellite attitude control system comprises, according to the invention:

- as main actuators, two control moment gyros, the gimbal axes of which are parallel to each other and, typically, to the Z axis, it being possible for their angular momenta $\vec{H}_1$ and $\vec{H}_2$ to be independently oriented in all directions in the (X,Y) plane, as shown in figure 2a, and also a third actuator, advantageously at least one reaction wheel, used as a complement for

delivering torques, in both senses, out of the plane of the angular momenta of the control moment gyros (for example along the +Z and -Z directions), this other actuator being called in the rest of the description the Z-axis main actuator. Figure 3 shows schematically such an example of three main actuators.

In the embodiment shown in figure 3, which is of a type that can be used in particular on a satellite, the platform of which is shown schematically as 1, the Z-axis main actuator is a reaction wheel 2, with a variable rotation speed and controlled about the Z axis, for example the yaw axis of the satellite, whereas the two control moment gyros 3 and 4 each have a rotor, 5 and 6 respectively, which are driven so as to rotate at a controlled speed, which is usually constant but which can be variably controlled, about a rotation axis contained in the plane defined by the roll axis X and the pitch axis Y of the satellite, each rotor 5 and 6 being mounted so as to rotate on a steerable gimbal 7 and 8 respectively, by an electric motor 9 and 10 respectively, about a gimbal axis A_1 or A_2 respectively, which is parallel to the yaw axis Z. Thus, it will be understood that the angular momenta $\vec{H}_1$ and $\vec{H}_2$ of the control moment gyros 3 and 4 may be independently, and possibly simultaneously, oriented in all directions in the (X,Y) plane by rotation of the gimbals 7 and/or 8 about their respective gimbal axis A_1 or A_2, whereas the reaction wheel 2 delivers, complementarily, torques along the +Z and -Z directions lying outside the plane of the angular momenta $\vec{H}_1$ and $\vec{H}_2$ of the control moment gyros 3 and 4;

- as secondary actuators, the system also includes a set of actuators such as, for example, magnetic-torquers, jet actuators, steerable reflecting ailerons or tabs, or any other actuators necessarily used on board satellites for counteracting the cumulative effects of external disturbing torques always present in orbit;

- as sensors, the system also includes a set of external or inertial attitude sensors for reconstructing the attitude angles and angular velocities of the satellite with respect to a three-axis (geocentric or inertial or other) reference frame and thus for measuring the differences between these reconstructed attitudes and velocities and the desired attitude and velocity of the satellite along the three axes, and also sensors needed for control mechanisms of the main and secondary actuators (measurements of the speed of the wheel 2, measurements of the positions and rotation speeds of the gimbals 7 and 8 about the gimbal axes of the control moment gyros 3 and 4, etc.); and

- as computational means, the system further includes any type of computational member (microprocessor, computer, DSP, ASIC, FPGA, microcontroller, electronic circuit, etc.) for producing, in digital or analog form, or digital/analog hybrid form, signals representative of at least the following quantities: setpoint variables, such as the attitude angles (or quaternion) of the satellite, the angular velocity of the satellite, the inertial orientation of an axis fixed in the satellite reference frame, setpoints needed to perform satellite orientation maneuvers (setpoint trajectory in terms of attitude, speed, angular acceleration, temporal profiles of the torques or angular momenta needed to carry out the maneuver, etc.), estimation of the attitude and angular velocity of the satellite with respect to the three axes on the basis of the measurements of the sensors, commands to be sent to the main and secondary actuators, etc.

In the initialization phase of the system, advantageously starting from a configuration in which the angular momenta of the two control moment gyros are equal and opposite ($\alpha = 180°$), the secondary actuators, and optionally the third, Z-axis main actuator 2 when

it is skewed and can generate a angular momentum component in the (X,Y) plane, are operated, in parallel or sequentially, in order to generate an angular momentum in at least one direction in the (X,Y) plane for bringing, by negative feedback, simultaneously or sequentially, the pair of control moment gyros 3 and 4 into a configuration in which the angle α has a value sufficiently far from 180° without however being zero, the total angular momentum of the pair of control moment gyros thus being nonzero and opposed to the angular momentum generated by the secondary actuators.

The angular momentum skew ($\vec{H}_1 + \vec{H}_2 = \vec{H}$) of the pair of control moment gyros created by α different from 180° may be positioned advantageously (but not necessarily) along the normal to the orbital plane of the satellite, so as to limit the angular momentum transfer between the control moment gyros during the operational phase and during the tilting of the satellite.

For the same reason, the angular momentum skew ($\vec{H}$) of the pair of control moment gyros created by α different from 180° may advantageously be compensated for by the projection in the (X,Y) plane of the cumulative angular momentum for this purpose by the third main actuator (figure 2b), so that the cumulative total angular momentum produced by the actuators is zero.

To give an illustration, one particular redundant configuration based on two control moment gyros and a third main actuator consisting of two reaction wheels, implementing this method is shown in figure 2b, in which Z1 and Z2 denote the rotation axes of the two reaction wheels, which axes are skewed relative to Z, and $\vec{H}'_1, \vec{H}'_2$ are the angular momentum vectors, produced by the two reaction wheels by controlling their rotation speed in such a way as to compensate for the total angular momentum $\vec{H}$ of the pair of control moment gyros. Thus, if $\vec{H}_1 + \vec{H}'_2 = \vec{H}$ then $\vec{H}'_1 + \vec{H}'_2 = \vec{H}$.

Once positioned in this configuration in which α is nonzero and different from 180°, the pair of control moment gyros 3 and 4 may be used very simply to ensure controllability of the satellite along the X and Y axes, without having to vary the speed of any of the two rotors 5 and 6.

This is because, as shown in figure 4a, if β_1 is the angle between $\vec{H}_1$ and the X axis and β_2 is the angle between $\vec{H}_2$ and the X axis, by varying β_1 and β_2 by the same small angle $\Delta\beta = \Delta\beta_1 = \Delta\beta_2$ over the time ΔT, α therefore remaining constant, the effect obtained is to rotate the total angular momentum $\vec{H} = \vec{H}_1 + \vec{H}_2$ of the two control moment gyros 3 and 4 about the Z axis, thus creating a torque $\Delta H/\Delta T$ normal to $\vec{H}$ (at small angles).

Moreover, by varying α (from α to α'), while keeping orientation of the bisector of the angle of the angular momenta $\vec{H}_1$ and $\vec{H}_2$ constant (which moments become $\vec{H}'_1$ and $\vec{H}'_2$ for α'), in other words by rotating the control moment gyros 3 and 4 in such a way that $\Delta\beta_1 = -\Delta\beta_2$, as shown in figure 4b, the total angular momentum $\vec{H} = \vec{H}_1 + \vec{H}_2$, which becomes $\vec{H}' = \vec{H}_1' + \vec{H}_2'$, varies in norm (from ΔH) but not in direction, thus creating a torque in the direction of H or of -H.

In total, by independently varying both β_1 and β_2 by a suitable amount $\Delta\beta_1$ and $\Delta\beta_2$, any torque can be very rapidly created along any direction in the (X,Y) plane, thereby ensuring complete (X,Y) controllability, and also its quasi-decoupling with control about the Z axis at small angles and/or low angular velocities, provided by the Z-axis main actuator.

In this way, the satellite attitude control system uses the main actuators (the pair of control moment gyros 3 and 4 and the third main actuator 2) as nominal control means.

In fine (small-angle) pointing mode, on the basis of observed differences between setpoint variables (attitude, angular velocity, pointing of a reference axis, etc) and estimated variables, the computational member generates commands to be sent to these actuators in order to generate the torques needed to correct for these differences. The commands sent may be of various types, digital or analog, and pertain to various physical variables, such as for example the current to be injected into the motors such as 9 and 10 for the gimbal shafts and for the wheel or wheels such as 2 and the rotors 5 and 6, the absolute or relative position of the gimbals such as 7 and 8 in rotation about the gimbal axes, the gimbal rotation speed, the rotation speed of the wheel or wheels 2 and the rotors 5 and 6, etc. Their effect at small angles is always to create small torques about the X, Y, Z axes, allowing the satellite to be stabilized around the setpoint variables.

Advantageously, the variation in the angles β_1 and β_2 is calculated and applied so as to accomplish, alone and in totality, the desired servocontrol along the X and Y axes, and to do so using the abovementioned principle elements, which a person skilled in the art can easily use to define the precise algorithms to be implemented. The Z-axis main actuator (for example the reaction wheel 2 of figure 3) is used to accomplish, alone and in totality, the Z-axis servocontrol, and to do so in a manifestly independent manner at small angles.

The method of the invention makes it possible to install a system momentum offloading strategy. This is because, owing to the effect of the external disturbing torques, which act continually and cumulatively, the total angular momentum of the system of main actuators (control moment gyros 3 and 4 and wheel 2) does not cease to increase: the angular momentum of the Z-axis

third actuator (in the case of at least one reaction wheel 2) will have a tendency to increase up to saturation, and the pair of gyroscopic actuators 3 and 4 will have a tendency to be aligned in the $\alpha = 0°$ position, possibly passing via the undesirable $\alpha = 180°$ configuration. In this case, the system becomes uncontrollable. It is therefore necessary to limit the excursions of the speed of the wheel 2 and the range of variations of the angle α within acceptable limits (the specified angular range including neither 0° nor 180° in the case of α) that depend on the precise design of the system. To achieve this offloading effectively, secondary actuators are used, either in open loop, for example by estimating the orbital disturbing torques and compensating for them, or in closed loop, or by combining these two solutions. To give an example, the control system sends commands to these secondary actuators, which have the effect of creating a variation in the angular momentum in the same sense as its observed increase, while still maintaining, of course, the attitude setpoints at their nominal value. In reaction to these effects, the system of main actuators 2, 3 and 4 can but produce its own accumulated angular momentum, thereby moving the actuators (control moment gyros 3 and 4 and third actuator 2) away from their saturation zone.

Starting from an initial configuration of the control moment gyros 3 and 4, in which α may be close to 180°, the large-angle maneuvers are advantageously carried out by implementing a method similar to that described in FR 2 786 283, to which the reader may refer for any details on this subject, which document is incorporated in the present specification by way of reference.

To carry out rapid tilting maneuvers, the method advantageously includes the following steps in which:

- a setpoint configuration for the pair of control moment gyros, away from the singular configurations,

that is to say from the configurations for which the angle α is zero or equal to 180°, and possibly a temporal Z-axis angular momentum profile that has to be performed by the third, Z-axis main actuator are determined from the initial and final conditions of the satellite in terms of attitude angles, angular velocity and time, in such a way that the angular momentum exchange, over an imposed duration, between the satellite, the two control moment gyros brought into said setpoint configuration and the Z-axis third actuator, brings about the desired attitude maneuver; and

- the orientation of the gimbal of each of the control moment gyros is brought, simultaneously and possibly independently, into its orientation in the setpoint configuration thanks to an angular position setpoint sent, in open loop, into a local servomechanism for controlling the angular position of the gimbals; and

- the Z-axis angular momentum profile is generated, simultaneously and possibly independently, using the third, Z-axis main actuator, advantageously at least one reaction wheel, the rotation speed of which will consequently be varied.

In this method, the reorientation of the control moment gyros in their setpoint configuration will advantageously be accomplished very rapidly, and may consequently pass transiently through singular positions (α = 0° or 180°) without it being prejudicial to the proper behavior of attitude control.

Since the setpoint configuration of the control moment gyros is chosen to be a nonsingular position, the system remains controllable in this configuration, so that, advantageously, on the basis of differences observed in the generation of the maneuver profile relative to a predefined setpoint profile, closed-loop

commands are added to the open-loop setpoints sent to the main actuators so as to reduce said differences.

The setpoint configuration away from the singularities will be chosen according to the controllability reserves that the designer will wish to have in order to perform the closed-loop control during the maneuver. For a given tilting maneuver, it will always be possible to increase this distance from the singularities by for example limiting the maximum tilt speed.

In this maneuvering mode, the homing of the pair of control moment gyros onto its setpoint configuration advantageously takes place as rapidly as possible (the sole limitations being the capacity of the motors for the gimbal shafts of the control moment gyros and the rigidity of said control moment gyros), so as to optimize the duration and implementation of the maneuver.

In this method, the rapid reorientation of the pair of control moment gyros into its setpoint configuration ensures essentially tilting of the satellite. The Z-axis actuator is used simply to manage, during the maneuver, the transfer of the initial angular momentum (at $t = t_0$) $\vec{H}^0 = \vec{H}_1{}^0 + \vec{H}_2{}^0$ of the pair of control moment gyros (due to the skew between the said control moment gyros) from the (X,Y) plane to the Z axis, completely or partly according to the maneuver, so that this angular momentum $\vec{H}^0$ remains inertial. Since the maneuver is known, it is easy to reduce the temporal profile of the angular momentum transfer to be produced with the Z-axis main actuator so that the coupling between the X, Y and Z axes is very easy to manage according to our method.

The three-axis attitude control system and method according to the invention thus make it possible, by

minimizing the number of main actuators of the control moment gyro type, and therefore allowing substantial savings in terms of weight, power, volume and cost, to control satellites for which agility essentially about two axes is required.